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AFRPL-TR-69-238

EVALUATION OF A HEAT-FLUX MEASUREMENT SYSTEM

J. R. ELLISON, LT, USAF
H. I. BINDER

TECHNICAL REPORT AFRPL-TR-69-238

NOVEMBER 1969

AIR FORCE ROCKET PROPULSION LABORATORY
AIR FORCE SYSTEMS COMMAND
UNITED STATES AIR FORCE
EDWARDS, CALIFORNIA

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AFRPL-TR-69-238

EVALUATION OF A HEAT-FLUX MEASUREMENT SYSTEM

J. R. Ellison, Lt, USAF

and

H. I. Binder

FOREWORD

This report was prepared jointly by the Motor Component Development Branch, Solid Rocket Division, Air Force Rocket Propulsion Laboratory (AFRPL), and the Instrument Research and Development Section of the Technical Support Division. The subject test was conducted at the AFRPL under Project 305903 AMG, Solid Rocket Hardware Design and Evaluation, with Lt John R. Ellison as Project Engineer. The nozzle was procured from the Aerotherm Corporation, Mountain View, California under Contract F04611-68-C-0086, with Mr. Harold I. Binder as Project Engineer. The Aerotherm Corporation Project Engineer was Mr. John W. Schaefer. This report describes the nozzle test which was conducted in August 1969.

This report has been reviewed and approved.

CHARLES R. COOKE
Chief, Solid Rocket Division
Air Force Rocket Propulsion Laboratory

ABSTRACT

A solid rocket nozzle instrumented with total and radiative heat-flux transducers was test-fired 13 August 1969 on the AFRPL 36-inch Char motor. The nozzle was a conventional heavyweight test configuration. The test objective was to measure the total and radiative heat-flux components incident on the nozzle ablative liners. Thermocouple instrumented plugs of ablative material comprised the total heat-flux measurement system. Plugs were placed in the entrance cap, throat, throat extension, and exit sections of the nozzle. A narrow-view angle radiometer provided radiative heat-flux measurements at the throat. A 6500°F flame temperature uncured propellant was used to provide a 1061-psig maximum chamber pressure for a 54-second total firing duration. A description of the heat-flux measurement system concept, nozzle design, motor preparation, performance, and posttest analysis is included in the report.

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SECTION I

INTRODUCTION

This report describes a total and radiative heat-flux measurement system and the results of the Air Force Rocket Propulsion Laboratory (AFRPL) test firing. The nozzle was procured from the Aerotherm Corporation under Contract F04611-68-C-0086.

The objective of this test was to demonstrate a technique for measuring total and radiant heat flux incident on nozzle ablative material during a rocket firing. The objective was achieved by: (1) designing and fabricating a 2.3-inch-throat-diameter solid rocket nozzle and instrumenting it with heat-flux indicators; (2) the evaluation and selection of propellant ballistic properties to obtain 6500°F flame temperature, 1061 psig maximum chamber pressure, 48 sec effective firing duration; and (3) posttest evaluation of the test data.

The gas generator used for this test was the AFRPL 36-inch-diameter Char motor. The motor utilized silica-filled Buna-n rubber insulation that was reconditioned by cleaning before firing. The motor fired vertically upward using uncured solid propellant in an end-burner configuration.

An uncured propellant, APG-112D, produced by the Atlantic Research Corporation, was selected for the test. The propellant was a 27% aluminum, 95% total solids formulation, with a nominal flame temperature of 6500°F. Desired test parameters were a 60-second firing duration and 740-psia chamber pressure. The firing was conducted 13 August 1969. Detailed propellant exhaust properties are shown in Table I.

TABLE I. CHARACTERISTICS OF APG-112D PROPELLANT

	CHAMBER	EXHAUST
Specific Impulse (lb-sec/lb)	262.4	
Characteristic exhaust velocity (ft/sec)	5042	
Temperature (°K)	3894	2719
Specific Heat (cal/100g-°K)	44.26	42.96
Moles of gas (g moles/100g)	2.9833	2.7849
Pressure (psia)	1000	14.699
Combustion Products g moles/100 g		
C	0.0000	0.0000
H	0.2678	0.1057
O	0.0074	0.0003
N	0.001	0.0000
Cl	0.0648	0.0333
Al	0.0121	0.0003
CO	0.5403	0.5465
CO_2	0.0191	0.0128
CN	0.0000	0.0000
HCN	0.0000	0.0000
H_2	0.9790	1.1281
H_2O	0.2909	0.1758
HCl	0.3494	0.4904
O_2	0.0009	0.0000
OH	0.0492	0.0048
N_2	0.2659	0.2682
NO	0.0042	0.0002
NH	0.0003	0.0000
NH_2	0.0001	0.0000
NH_3	0.0000	0.0000
Cl_2	0.0001	0.0000
AlH	0.0017	0.0000
AlO	0.0045	0.0000
Al_2O	0.0023	0.0000
Al_2O_2	0.0003	0.0000
AlCl	0.1112	0.0168
$AlCl_2$	0.0051	0.0010
$AlCl_3$	0.0006	0.0002
AlOCl	0.0059	0.0004
C (solid)	0.0000	0.0000
Al_2O_3 (liquid)	0.4335	0.4973
Al_2O_3 (solid)	0.0000	0.0000
AlN	0.0000	0.0000

SECTION II

NOZZLE AND INSTRUMENTATION DESCRIPTION

A. NOZZLE ASSEMBLY

The test nozzle used for the heat-flux measurement demonstration was a conventional heavyweight configuration. The entrance cap, throat, and forward exit cone were fabricated from MX4926, a carbon cloth phenolic, and the aft exit cone liner was MX2600-96, a silica phenolic composite. Prefire throat diameter was 2. 3 inches, and the exit diameter was 6 inches. The exit cone half angle was 15 degrees and the exit expansion ratio was 5. 0. Conventional ablative component fabrication techniques and nozzle assembly procedures were utilized. The nozzle outer shell was machined from 1020 steel.

A cross section of the test nozzle and instrumentation is shown in Figure 1. More detailed views of the total flux plugs, radiometer, and radiometer installation are shown in Figures 2, 3, and 4, respectively. The Char motor performance was monitored by two 0 to 1500-psig-capacity strain-gage-type pressure transducers. All data was recorded on a digital system.

A brief description of the theory behind each measurement system follows.

B. TOTAL-FLUX SYSTEM

The primary measurement devices for the total-heat-flux determination were five thermocouples located below the flame surface of the instrumented plug. Thermocouples A, B, C, and D (Figure 2) were tungsten 5% rhenium/tungsten 26% rhenium, while E was chromel/alumel. The thermocouple selection was based on the expected temperature gradients and surface recession.

The plugs were 1/2 inch in diameter and composed of the same material as the rocket nozzle ablative liner. The total-heat-flux measurement system differed from conventional thermocouple plugs in that the thermocouples were located closer to the flame surface, the plug material was more extensively characterized as to high-temperature thermal properties, and the data reduction techniques were much more complex. Data recorded from these thermocouples provided a continuous history of the thermal gradients throughout the test firing. When this information was coupled with previously measured material thermal properties (conductivity, emissivity), it was possible to calculate the heat flux incident on the surface of the thermocouple plug (QIN). The development of data reduction techniques to calculate this total heat flux was one of the contract objectives and will be discussed in detail in the contractor's final report. The foundation of these calculations is the energy balance relation. The energy balance which exists between the exhaust gases and the nozzle wall can be written as:

QIN = Qcond + Qrad out + Qchem

where:

QIN	=	heat flux at the material surface and into the component (convection, radiation)
Qcond	=	conduction heat flux into the material at the surface
Qrad out	=	radiation heat flux from the surface
Qchem	=	energy term which accounts for the chemical reactions, absorption or production, occurring at the material surface (details of this calculation are in Reference 1).

C. RADIATIVE-HEAT-FLUX MEASUREMENT SYSTEM

The primary sensing instrument was a narrow-view angle radiometer which employed a Gardon-type flux gage. The temperature difference between the center of the constantan foil sensor and the copper heat-sink body resulted in a varying thermocouple output, an indicator of heat flux. A nitrogen purge gas was utilized to provide convective cooling of the sensor body and the lens. The lens was necessary to shield the foil disk

from convective heat transfer. Since convective effects were eliminated, the Gardon-type heat-flux gage measured only the radiative heat flux. The measurement system was placed in a cavity below the initial surface of the nozzle, and was never directly exposed to the combustion products. The radiometer was located below the nozzle surface to prevent damage as the throat eroded. As a result, continuous radiative flux measurements were obtained. The narrow-view angle restriction was required to eliminate the possibility of interference from the side walls of the cavity.

SECTION III

TEST DESCRIPTION

A. TEST PREPARATION

The test nozzle was received from Aerotherm Corporation and installed on the 36-inch Char motor aft closure. An illustration of the nozzle installed on the closure is shown in Figure 5. The entrance cone was faired into the contour of the closure insulation with V-61 as shown in Figure 6. V-61 is a pottable insulation which can be cured at ambient conditions. Forty-eight hours prior to the scheduled motor loading, the chamber insulation was coated with Hycar polymer. The polymer provides a wetting agent that is compatible with the insulation and the propellant, thus preventing flame propagation down the propellant to insulation interface. The polymer was reapplied immediately before the propellant casting operation to insure that no 'ry areas existed. Vacuum casting fixtures were attached to the motor chamber and approximately 1100 pounds of propellant were cast into the motor. A combination of four $BKNO_3$ pellet bag igniters (see Figure 7) was installed 6 inches above the propellant surface. Concurrent with the igniter installation, the aft closure was installed. The multiple igniters were required because of the large motor free volume resulting from the use of a low-burn-rate propellant. The gaseous nitrogen (GN_2) purge system was regulated for a line pressure of 1300 psig and was connected to the radiometer purge system. An aerial view of the complete Char motor and test facility is shown in Figure 8.

B. MOTOR AND NOZZLE PERFORMANCE

The motor was fired on 12 August 1969, and a sharply regressive chamber pressure-versus-time trace (Figure 9) was obtained. Maximum chamber pressure was 1061 psig and average pressure was 580 psig. The actual test conditions were somewhat more severe than those desired as is shown in the following table.

Nozzle Design Conditions

Maximum Chamber Pressure	725 psig
Chamber Pressure at Tail-Off	160 psig
Total Duration	60 sec

Actual Test Conditions

Ma: imum Chamber Pressure	1061 psig
Chamber Pressure at Tail-Off	320 psig
Total Duration	54 sec
[1]Effective Duration	48 sec

The high chamber pressure was a result of an increased burn surface area. The reconditioning of the chamber insulation between previous firing removed e ough material to increase the burn surface beyond the desired 36 inches for which the nozzle was sized. The pronounced regressivity was expected since the carbon phenolic ablative throat had a very high surface recession rate.

No local perturbations or gouging were apparent in the vicinity of the total-flux plugs or the radiometer cavity. The silica phenolic exit extension performed very poorly as shown in Figure 10. Erosion rates were much higher than those of the carbon phenolic in spite of the higher area ratio location. A summary of the throat surface recession results in presented in the following table. A photograph of the throat region is shown in Figure 11.

[1]Effective duration is defined as the time from 75% of maximum pressure during rise to the point on the pressure trace which lies on a line bisecting the angle formed by the tangents to the curve prior to and immediately after tail-off initiation.

Measured Surface Recession

Average	0.47 inches
Maximum/Minimum	0.51/0.40 inches

Calculated Surface Recession (CMA Computer Program)

Actual Test Condition	0.46 inches
Nozzle Design Condition	0.36 inches

The general performance of the test nozzle is shown in the erosion profile, Figure 20.

Additional motor preparation and performance data, and nozzle performance data, are found in Table II.

TABLE II. MOTOR PERFORMANCE DATA SHEET

TEST TITLE HEAT FLUX SENSOR EVALUATION	
TEST NUMBER 40-031	TEST DATE 13 August 1969
Propellant Formulation - APG 112D	Strand Burn Rate - $.28\left(\frac{Pc}{800}\right)^{.67}$ in/sec
Theoretical Flame Temperature - 6500°F	7-Inch Motor Burn Rate - $.33\left(\frac{Pc}{800}\right)^{.67}$ in/sec
Prefire Throat Diameter - 2.30 inches	Predicted Max Pc - 1090 psig
Postfire Throat Diameter - 3.2 inches (nominal)	Actual Max Pc - 1060 psig
Prefire Motor Diameter - 38-1/8 inches	Ambient Temperature - 78°F
Postfire Motor Diameter - 38-7/16 inches	Ambient Pressure - 13.6 psia
As-Poured Propellant Depth - 15.0 inches	Cast to Fire Time Interval - 24 hr
Prefire Propellant Depth - 15.0 inches	Average Chamber Pressure - 580 psig
Prefire Propellant Temperature - 74°F	Casting Method - Vacuum
Propellant Weight - Nominal 1100 lbs	Average Throat Erosion Rate - 9.4 mil/sec

SECTION IV

POSTTEST ANALYSIS

A. INSTRUMENTATION PERFORMANCE

A qualitative assessment of the instrumentation performance is listed below, with additional discussion following.

Heat-Flux System	Station	Location[2]	Quality of Data	Comments
Total	1	$A/A_* = -2.0$	OK	Data in Figure 12
Total	2	$A/A_* = 1.0$	OK except	Erratic response of 3 thermocouples; valid results for complete firing expected however; data in Figure 13
Total	3	$A/A_* = 2.0$	OK	Data in Figure 14
Total	4	$A/A_* = 4.0$	OK	Data in Figure 15
Radiation	2	$A/A_* = 1.0$	OK to 3 sec	Apparent loss of nitrogen purge gas after 3 seconds; data in Figure 16

In the total-flux system at the throat, the three deepest thermocouples exhibted an erratic response. Sufficient data are available, however, to provide a valid definition of the flux at this location through the complete test firing. The reason for this erratic response has not been defined. None of the other plugs exhibited this type of behavior.

[2] A/A_* is defined as local area divided by throat area, and fixes the axial location. Negative numbers indicate subsonic locations.

In the radiative-flux system, a decrease in the required nitrogen purge gas flow apparently occurred at 3 seconds into the firing followed by a complete loss of this flow at 15 seconds. The failure at 15 seconds was due to the thermal decomposition of the supply hose caused by the high-radiation flux from the exhaust plume. The explanation for the event at 3 seconds is not so clear-cut, however. Several explanations are possible as outlined below:

1. The unexpected high chamber pressure resulted in a decrease in the design purge gas flow and therefore a thermal failure of the radiometer view restrictor and sapphire window (Figure 3). The nitrogen purge gas flow is controlled by a sonic orifice so that the variations in local static pressure (or chamber pressure) do not influence the flow rate. This requires a supply pressure slightly more than twice that of the maximum local static pressure (to also account for the pressure drop downstream of the sonic orifice). The maximum local static pressure was 620 psia, which therefore indicated a nitrogen purge gas supply pressure slightly greater than 1240 psia was required. The actual supply pressure was 1250 psia, indicating the potential for a decrease in the purge gas flow.

2. The pyrolysis products of the damaged hose plugged up the sonic orifice early in the test firing and thus caused radiometer failure. The hose failed at 15 seconds into the firing but probably started pyrolyzing before that time. These pyrolysis products could have condensed on the orifice, plugging it partially or completely. Postfiring inspection revealed that the orifice was almost completely plugged after the firing but this could have been due to the reverse flow after hose failure.

3. Dirt in the purge gas supply line plugged up the sonic orifice early in the firing and thus caused radiometer failure. The purge gas supply lines could have been dirty and some of this dirt could have been carried to the orifice.

It is not possible at this time to conclusively determine which of the foregoing possible explanations is correct.

B. DATA ANALYSIS

A preliminary data analysis was conducted. Examples of the results currently available from the reduction of the total-heat-flux measurement system data are presented in Figures 17 and 18. Figure 17 shows the measured in-depth temperature histories for the plug at $A/A_* = 2.0$ and the corresponding analytical fit of these results on which all subsequent data reduction is based. Note that the surface temperature and surface recession histories through the test (Figures 17 and 18, respectively) are also defined. The continuation of this data reduction process to define the total heat flux is now in progress for all measurement locations. Based on the excellent correlation of measured and predicted temperature data (e.g., Figure 17), the measured total heat fluxes are expected to approach within 20% of predictions.

The measured radiation-heat-flux history is presented in Figure 19. This measured radiation flux during the first 3 seconds is somewhat lower than was anticipated based on empirical calculation techniques, but is within the possible flux range. The empirical techniques for calculating radiative heat flux are heavily dependent on gas emissivity. In metalized propellants, the gas is composed of two phases: solid particles and gaseous combustion products. The uncertainty in the technique is due primarily to the impossibility of analytically assessing the contribution of the particle cloud. The quantitative comparison and interpretation of the results is affected by uncertainties in the conditions of the motor starting transient.

In summary, the total-heat-flux measurement systems provided reasonable data which will apparently allow reasonable total-heat-flux calculations. The radiative-heat-flux measurement system also provided reasonable data until an apparent decrease in, or failure of, the purge gas supply.

SECTION V

CONCLUSIONS

Based on the test results, the following conclusions were made:

1. The total-heat-flux measurement system provided reasonable data.

2. The radiative-heat-flux measurement system provided reasonable data for the first 3 seconds of the test.

3. The success of the radiative measurement system was limited by a combination of high motor pressure and a GN_2 purge line failure.

4. Acceptable correlation of theoretical and experimental temperature data was obtained.

5. The Char motor provided a more severe test environment than was anticipated during demonstration planning. The severe environment resulted from incompatible motor and nozzle throat diameters.

SECTION VI

RECOMMENDATIONS

Based on the test results, a more definitive evaluation of the radiative-heat-flux measurement system is desirable to increase the confidence level of the system performance. This could be accomplished by a second Char motor firing.

Operation procedures should be changed as follows:

1. The purge gas supply pressure should preferably be twice the anticipated maximum chamber pressure.

2. All purge lines in the vicinity of the Char motor should be insulated to prevent plume heating damage.

3. A filter should be installed as close to the sonic orifice as practicable to preclude the possibility of matter plugging the purge orifice.

4. The nozzle throat diameter should be increased to lower the motor operating pressure.

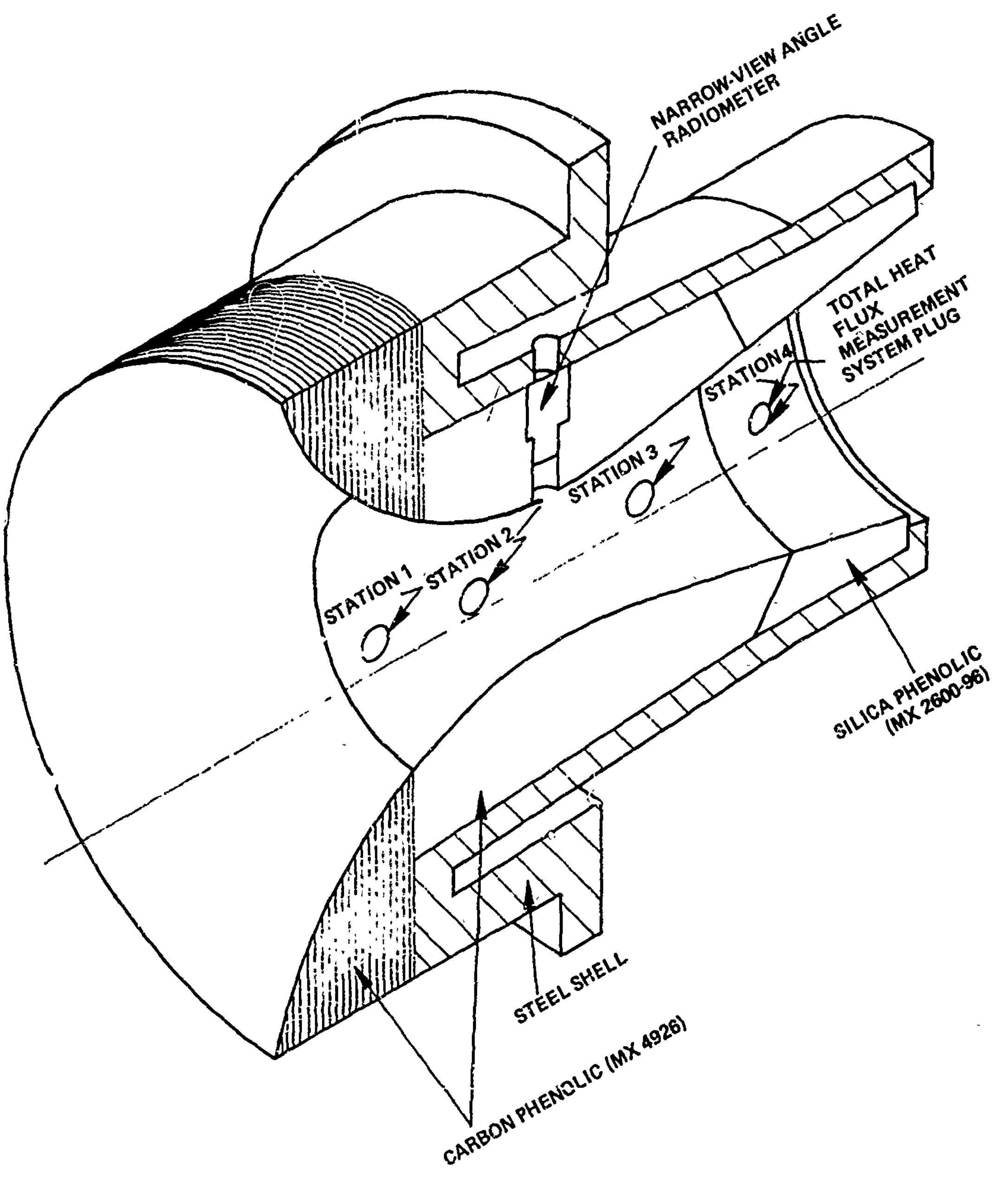

Figure 1. Char Motor Nozzle - Total and Radiative Heat-Flux Sensors

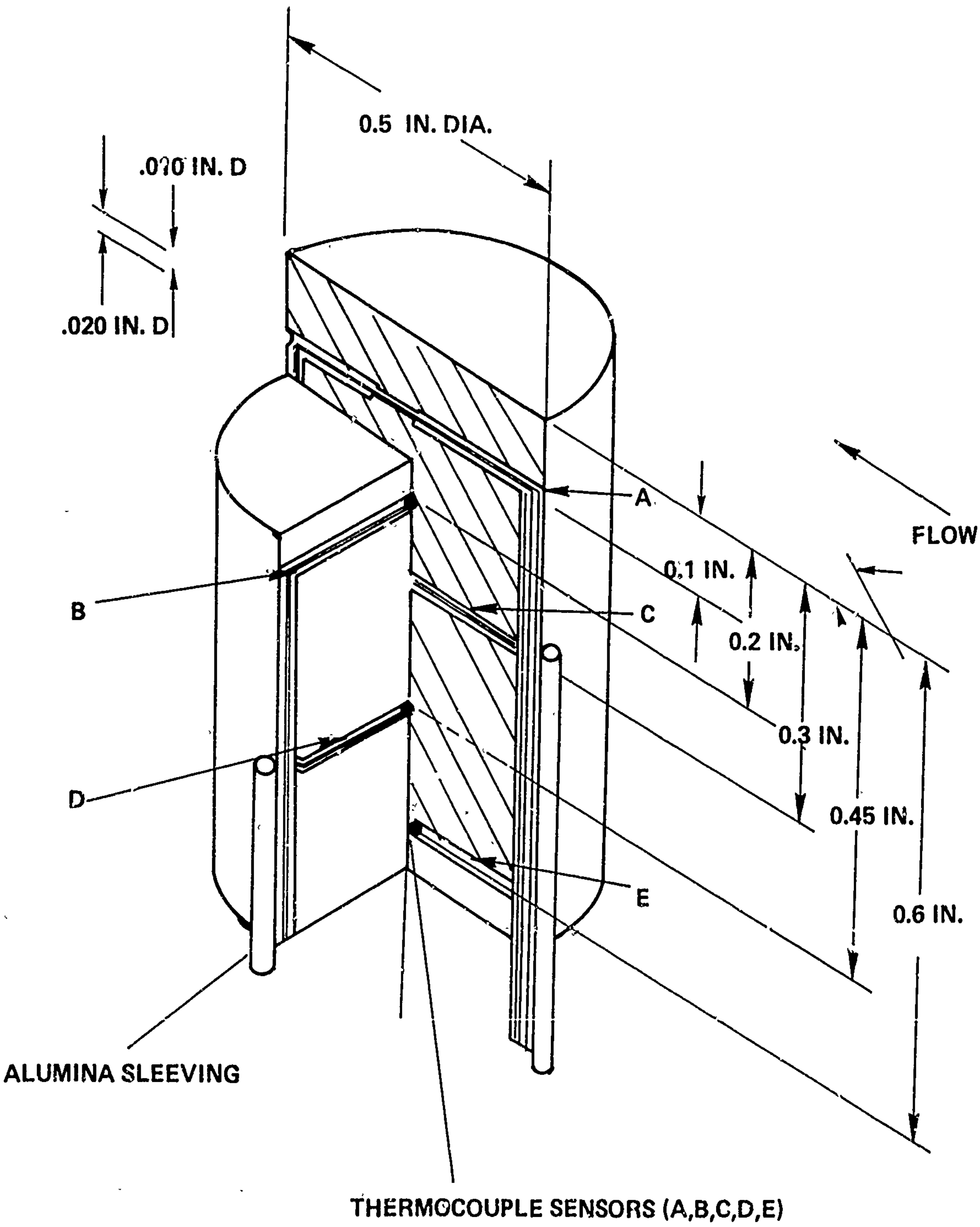

Figure 2. Total-Heat-Flux Measurement System Plug

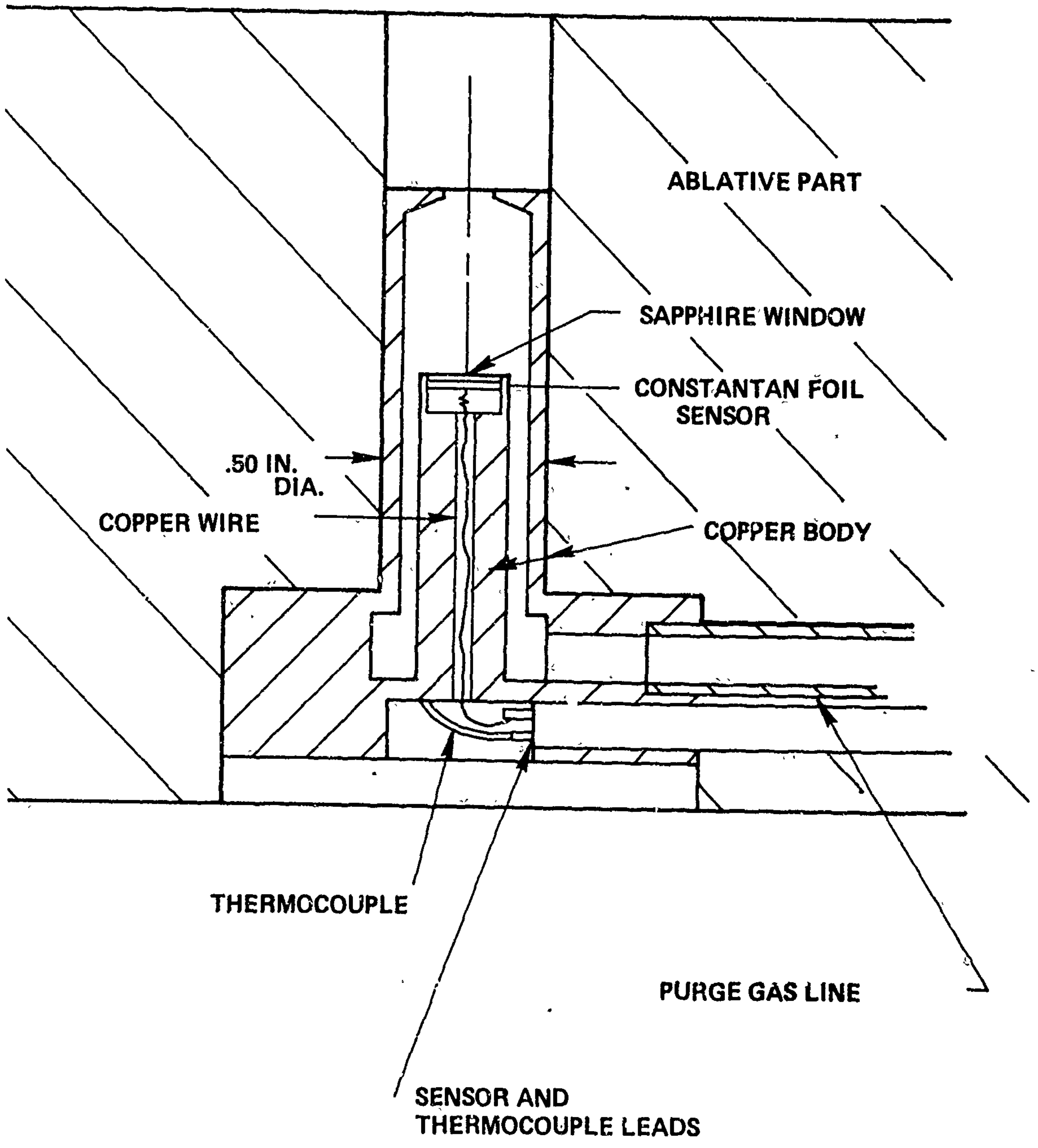

Figure 3. Schematic of Radiation-Heat-Flux Measurement System

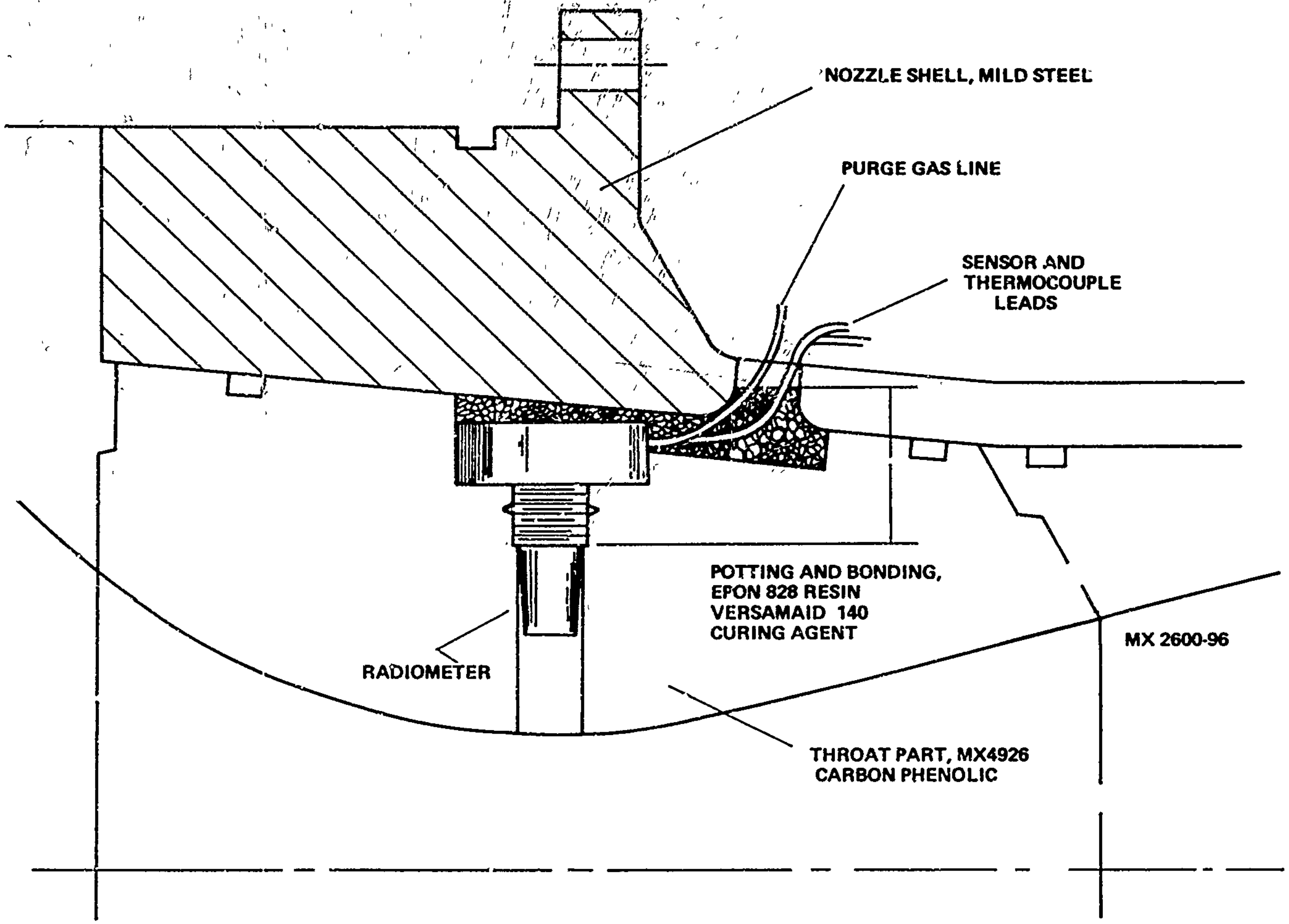

Figure 4. Radiation-Heat-Flux Measurement System Installation In Char Motor Nozzle

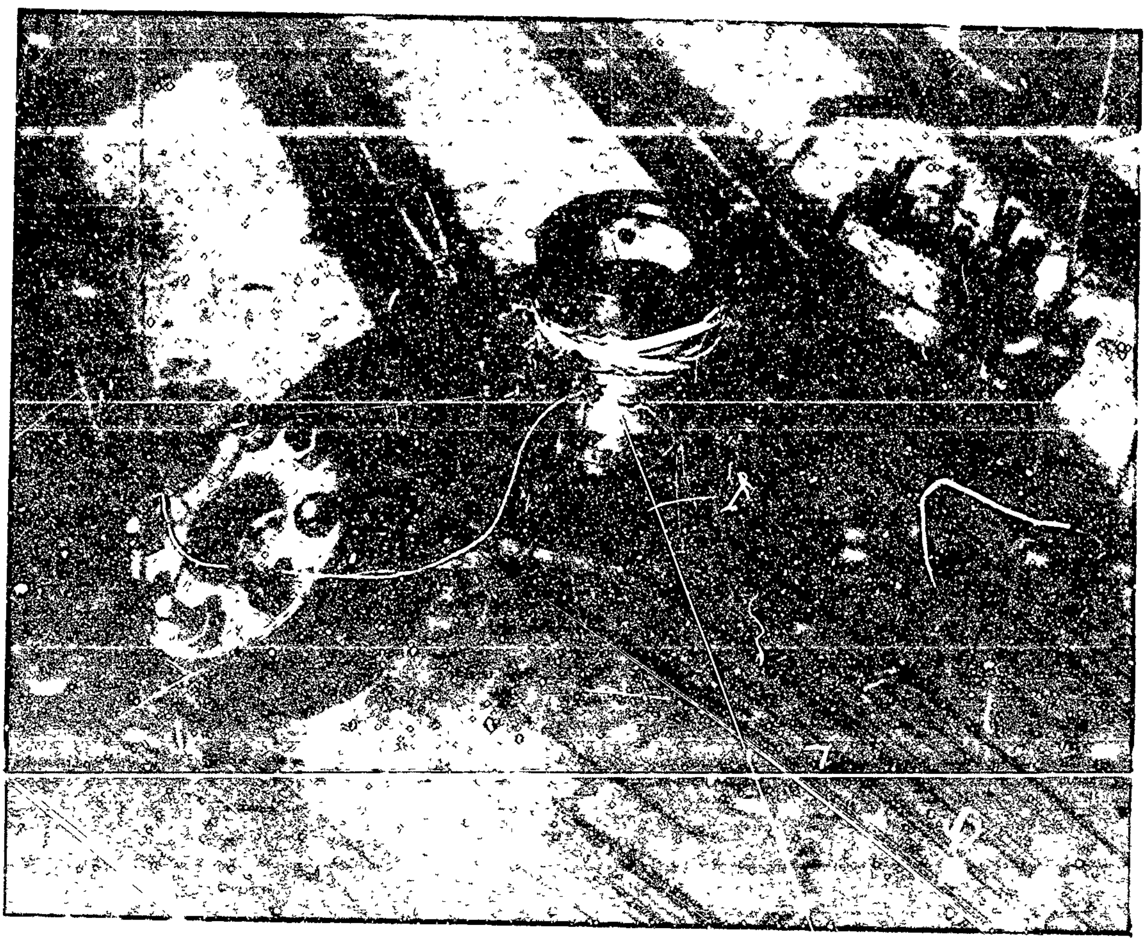

Figure 5. Prefire View of Test Nozzle and Aft Closure

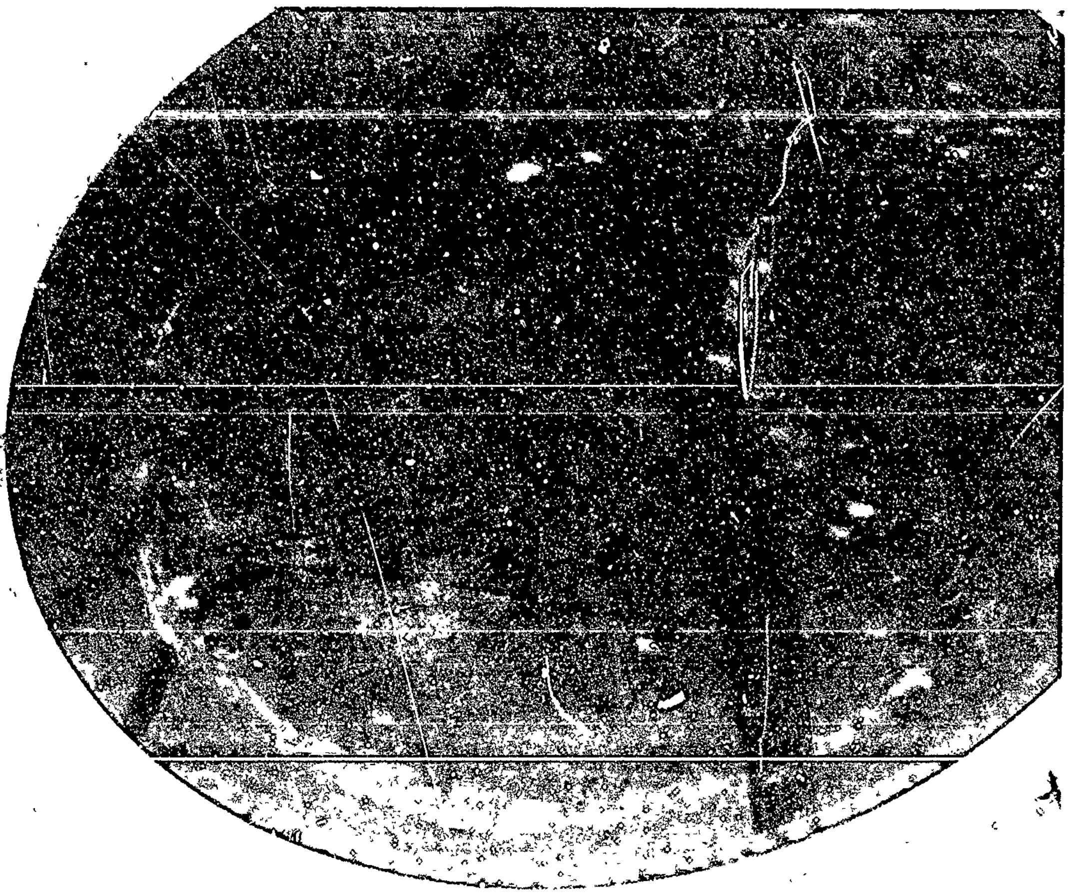

Figure 6. Prefire View of Inlet Section
With Fairing of V-61

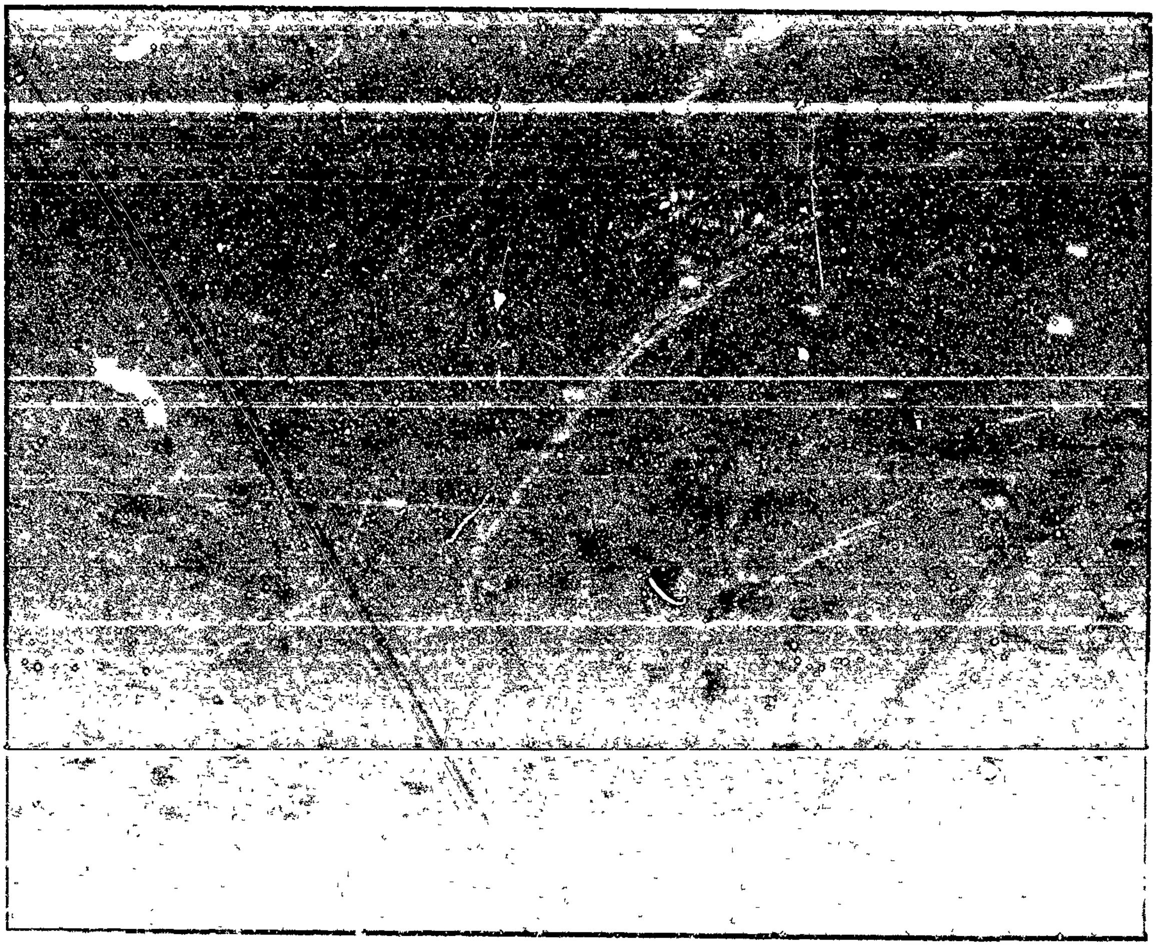

Figure 7. Multiple $BKNO_3$ Igniter Combination

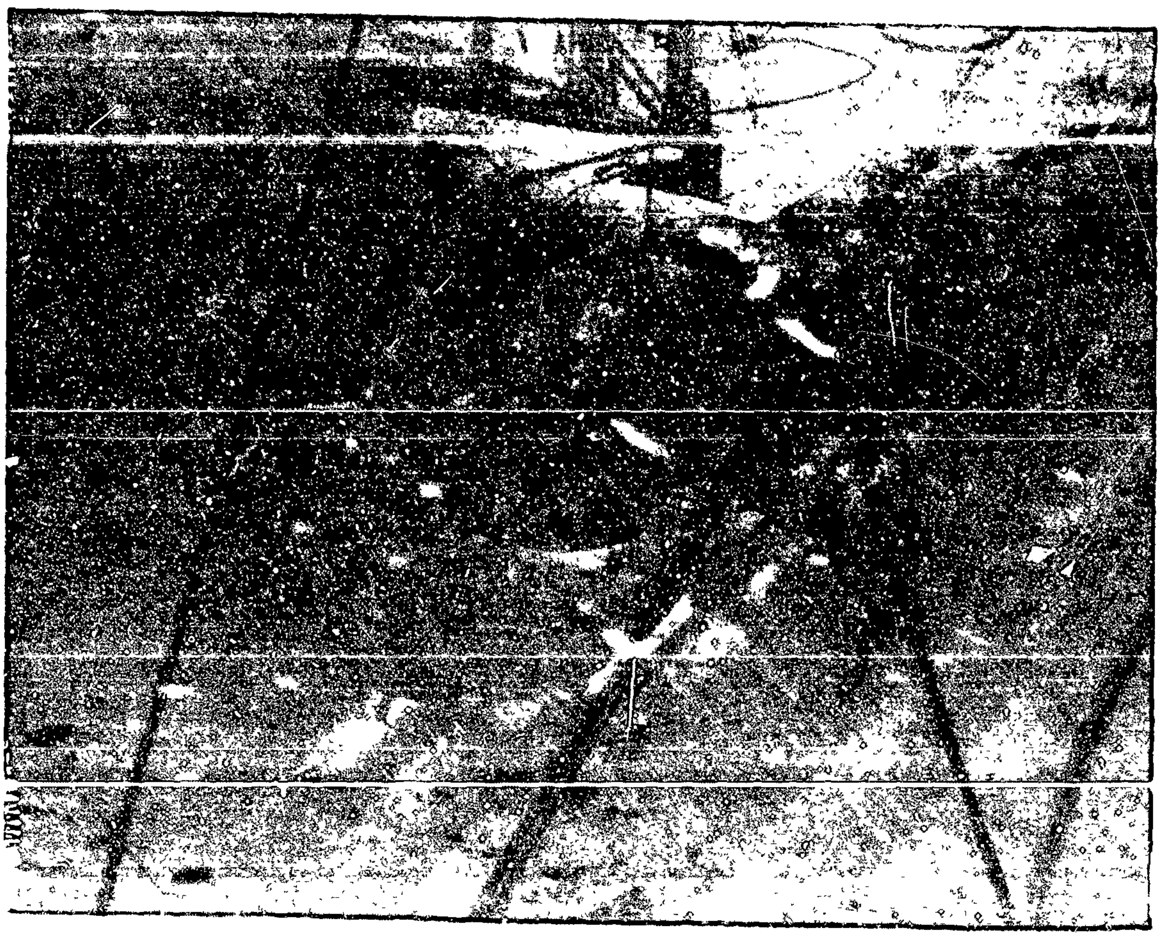

Figure 8. Aerial View of Test Facility

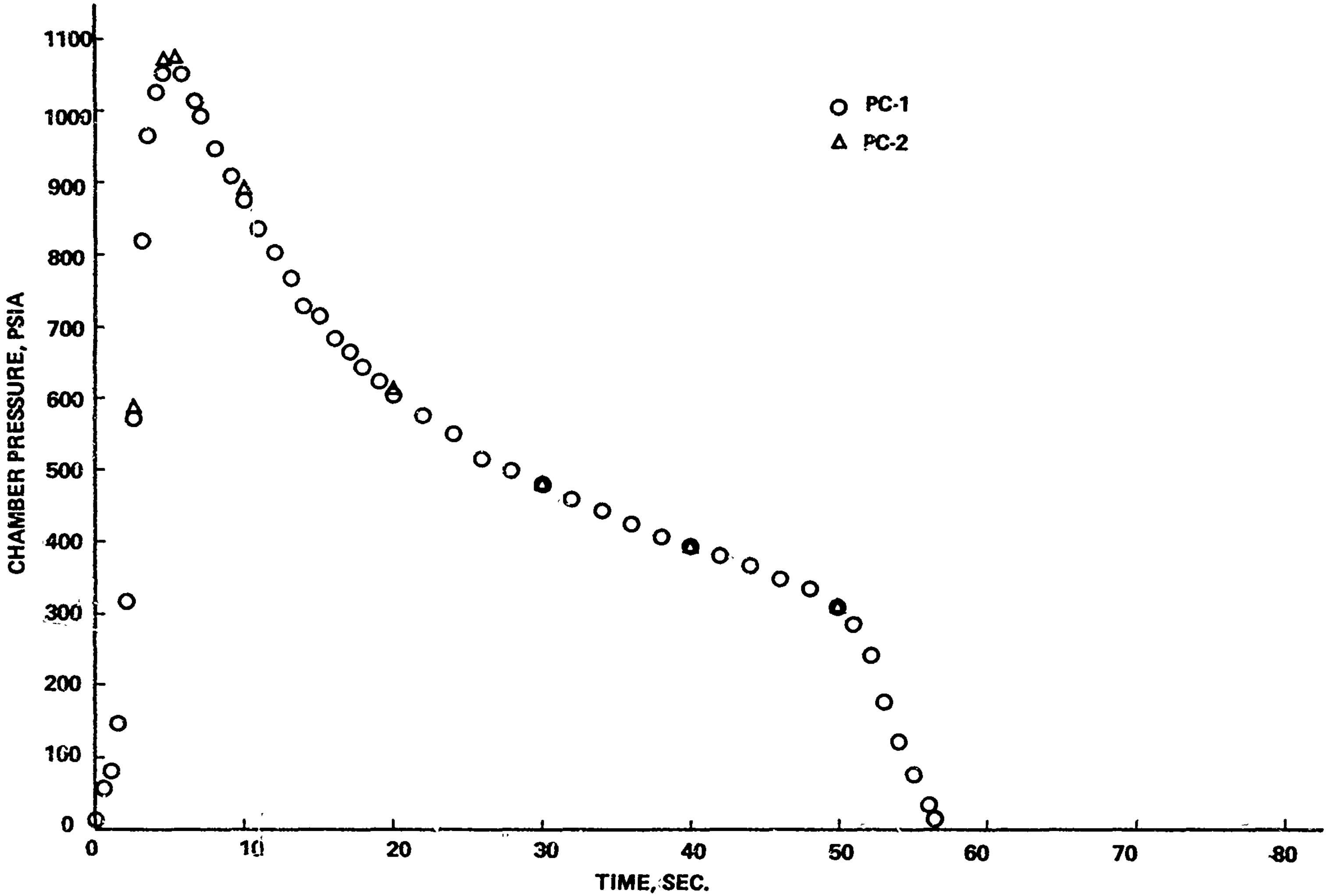

Figure 9. Chamber Pressure-versus-Time Trace

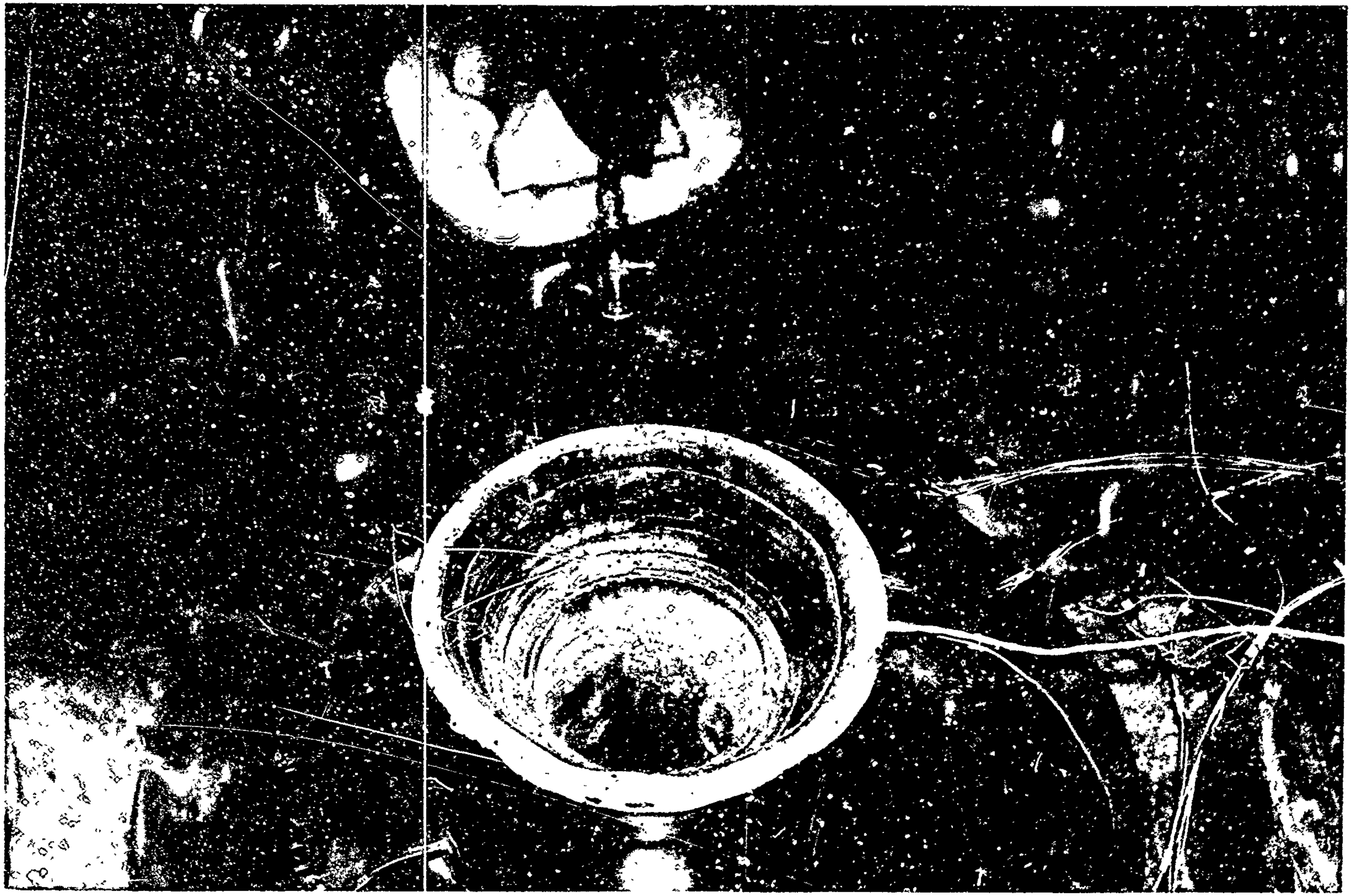

Figure 10. Nozzle Exit Cone, Postfire View

Figure 11. Throat Section of Nozzle

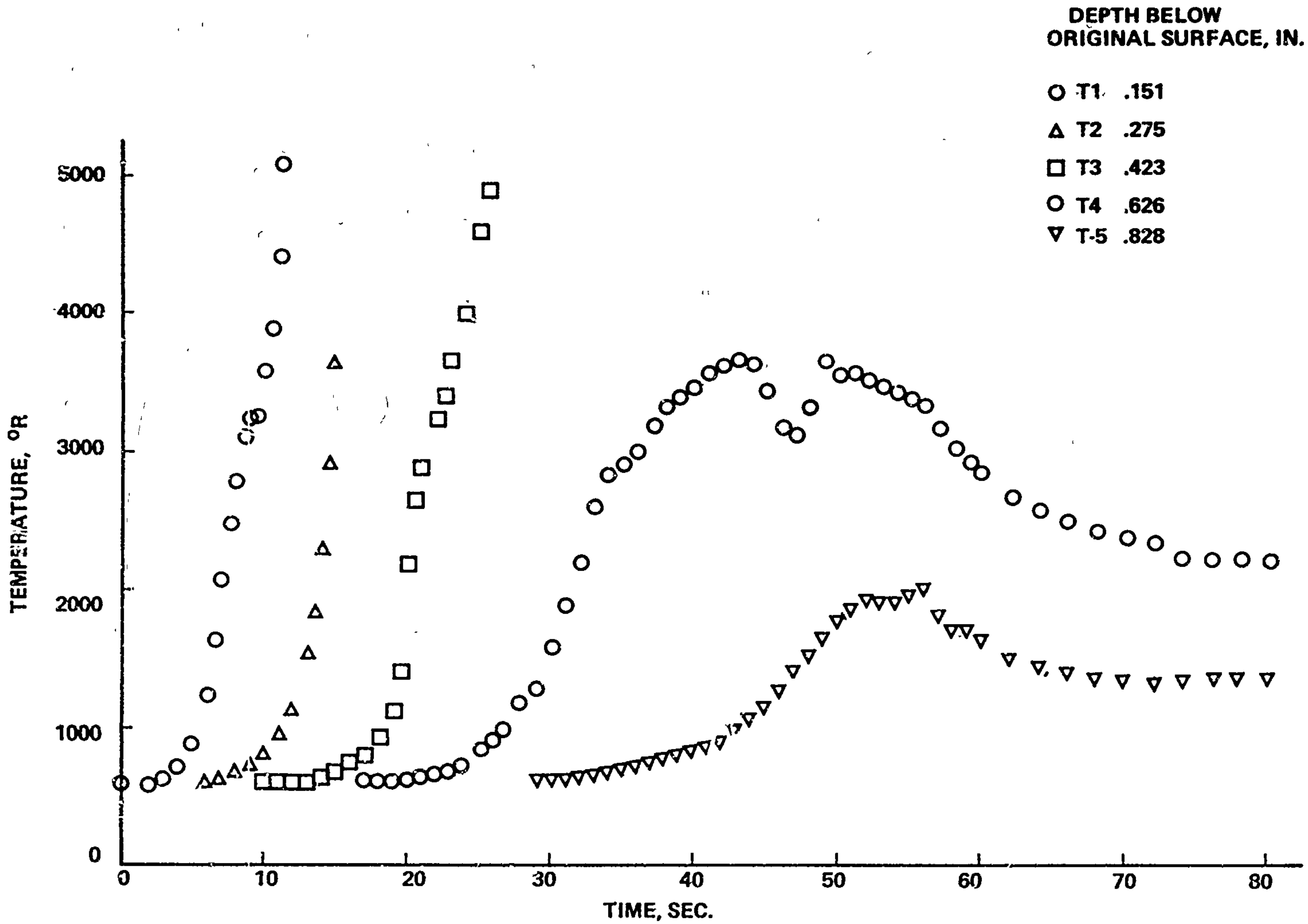

Figure 12. Measured Temperatures In Forward End of the MX4926, Station 1

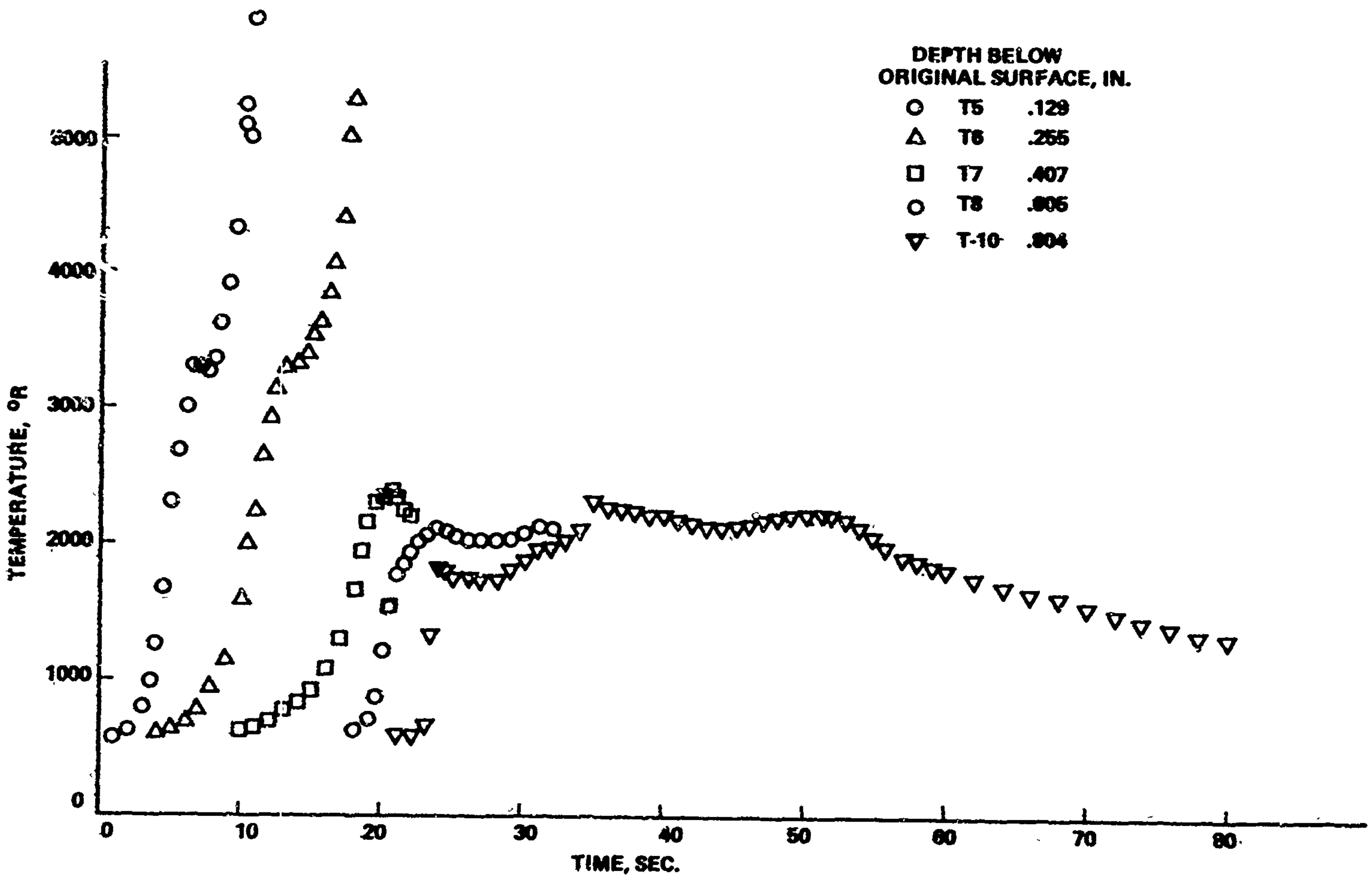

Figure 13. Measured Temperatures at the Throat of the MX4926, Station 2.

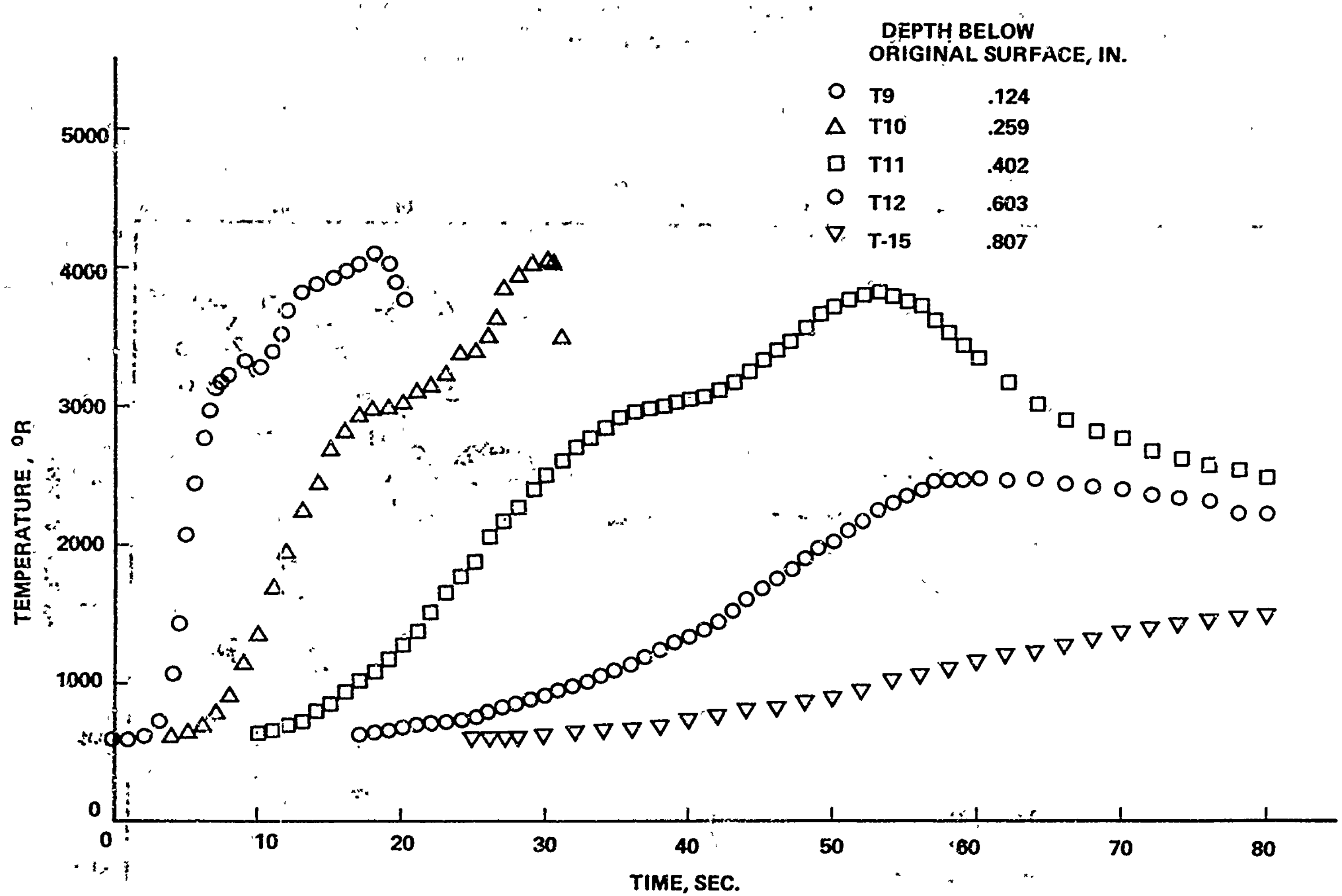

Figure 14. Measured Temperatures at the Downstream End of the MX4926, Station 3

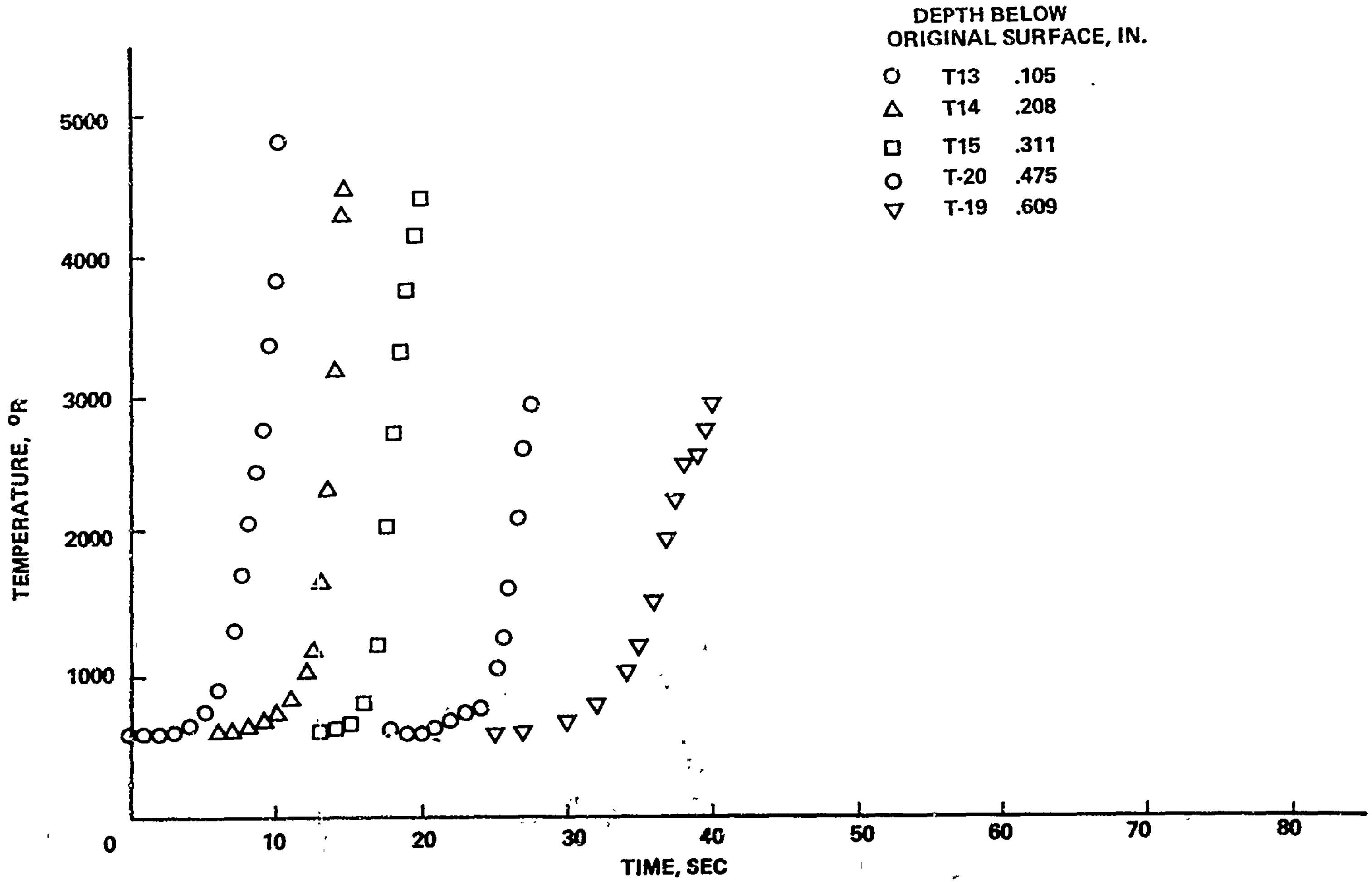

Figure 15. Measured Temperatures in the Exit Cone, MX2600, Station 4

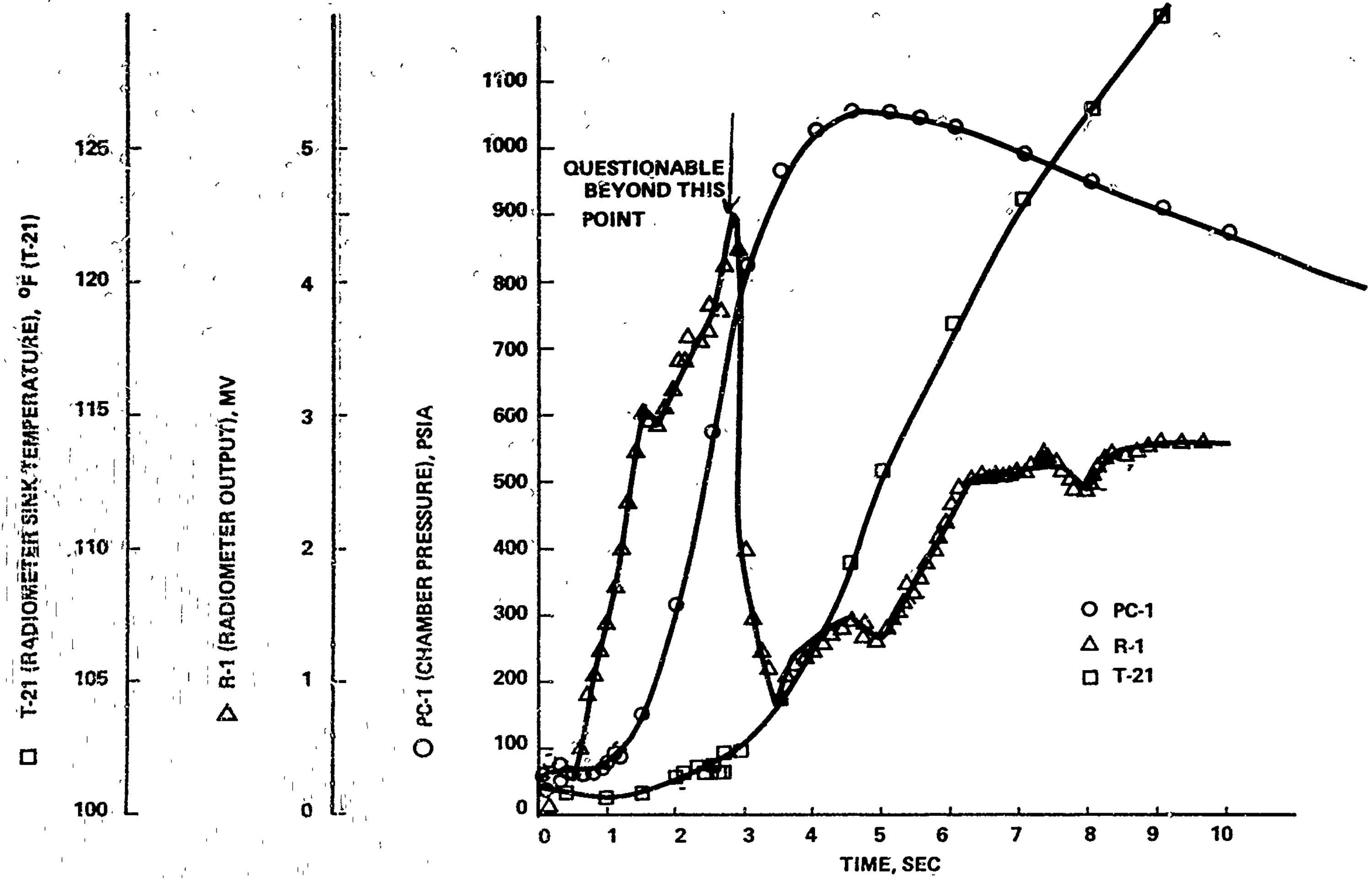

Figure 16. Radiometer Output and Associated Information, Station 2

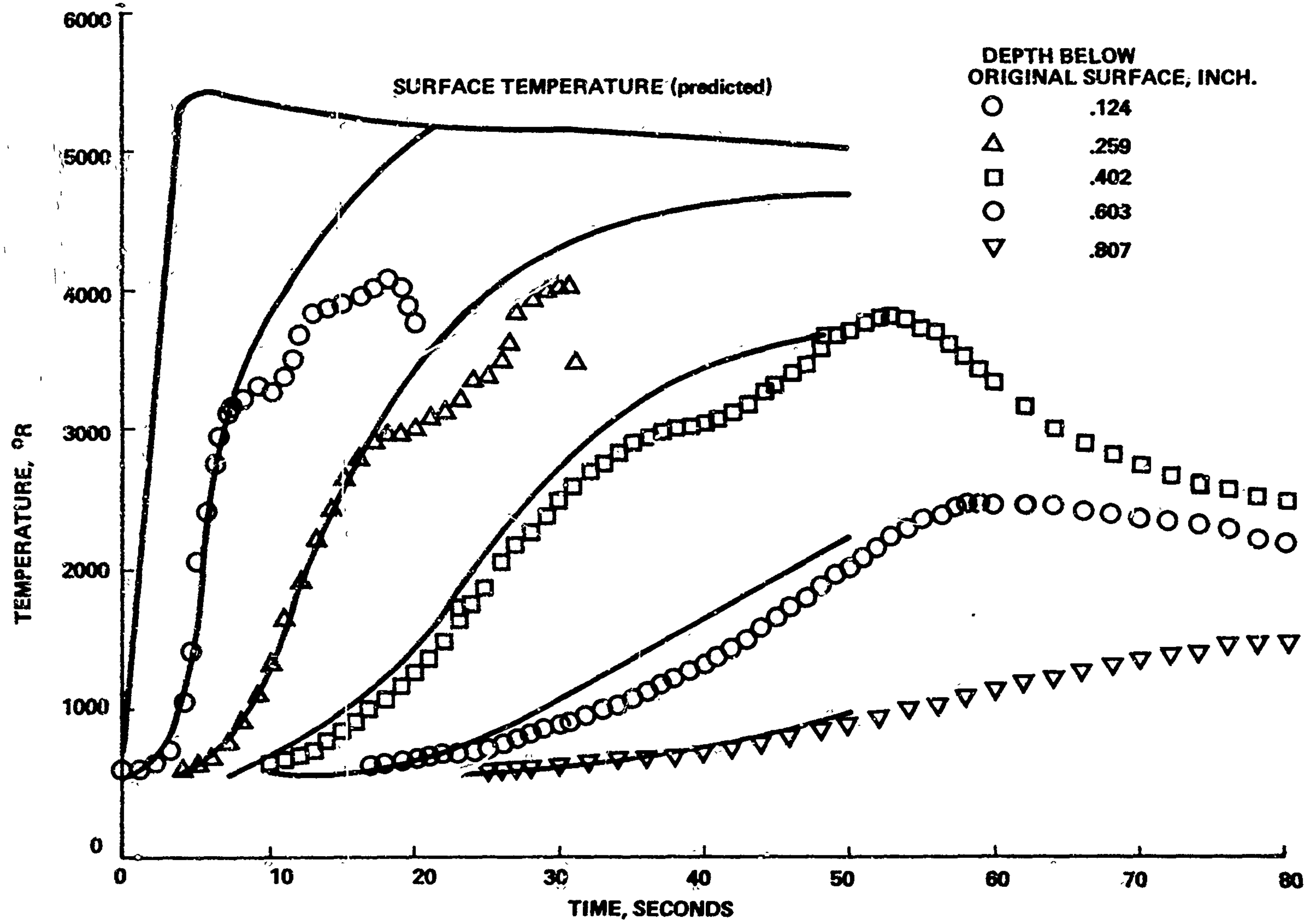

Figure 17. Measured and Predicted Temperatures for Station 3 in the MX 4926

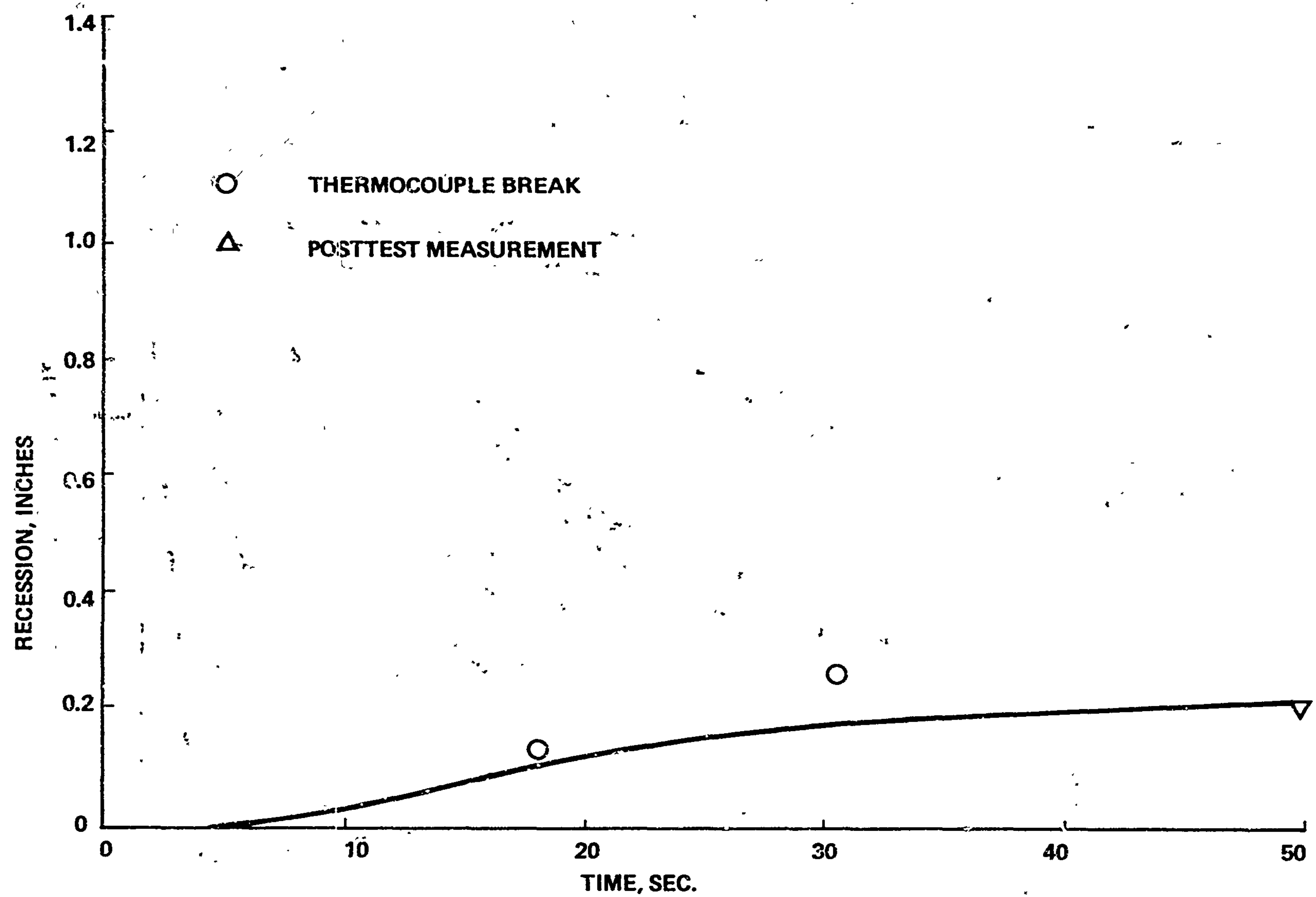

Figure 18. Surface Recession as a Function of Time For Station 3, MX4926

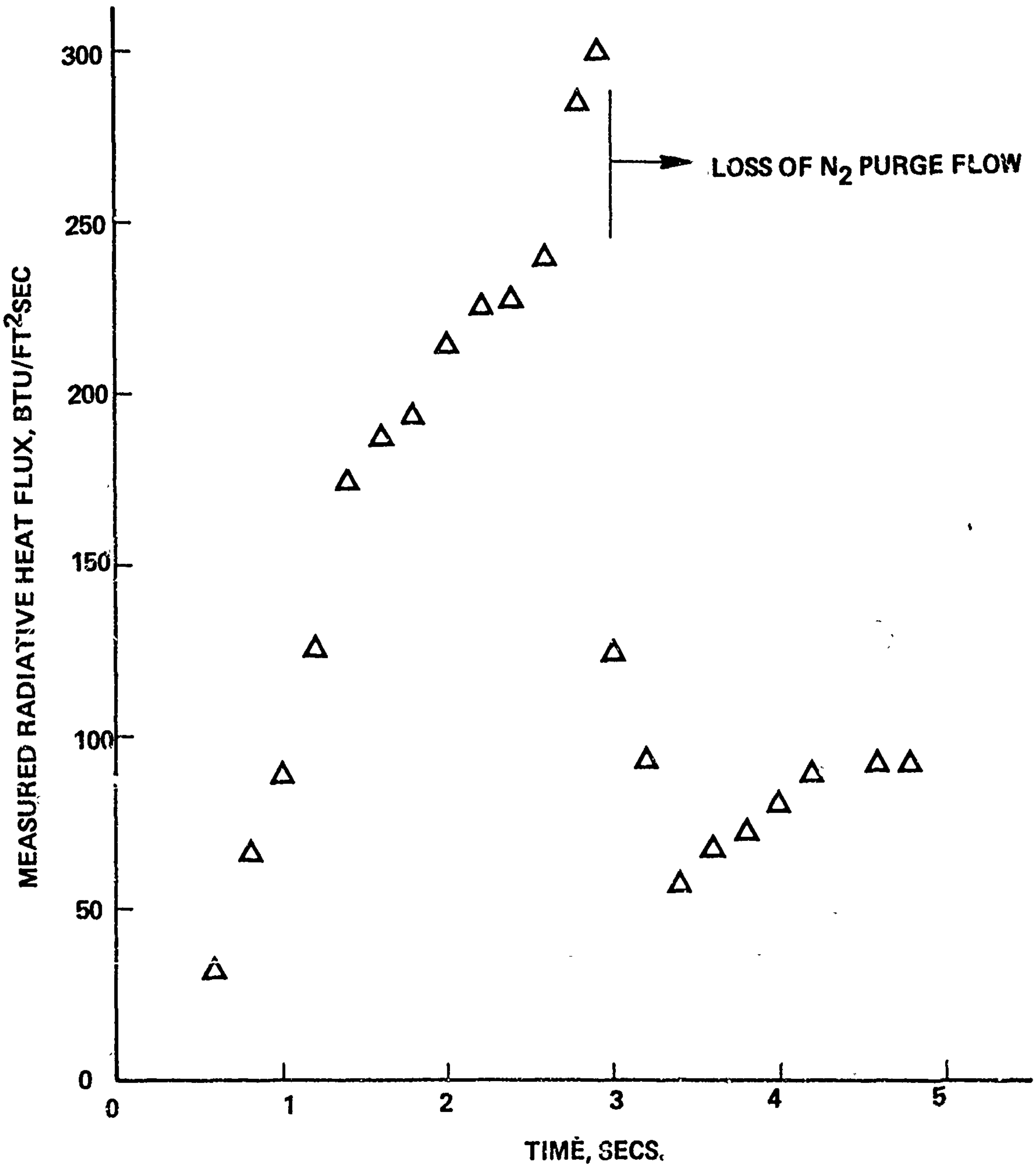

Figure 19. Measured Radiative Heat Flux at Throat Of Char Motor Nozzle

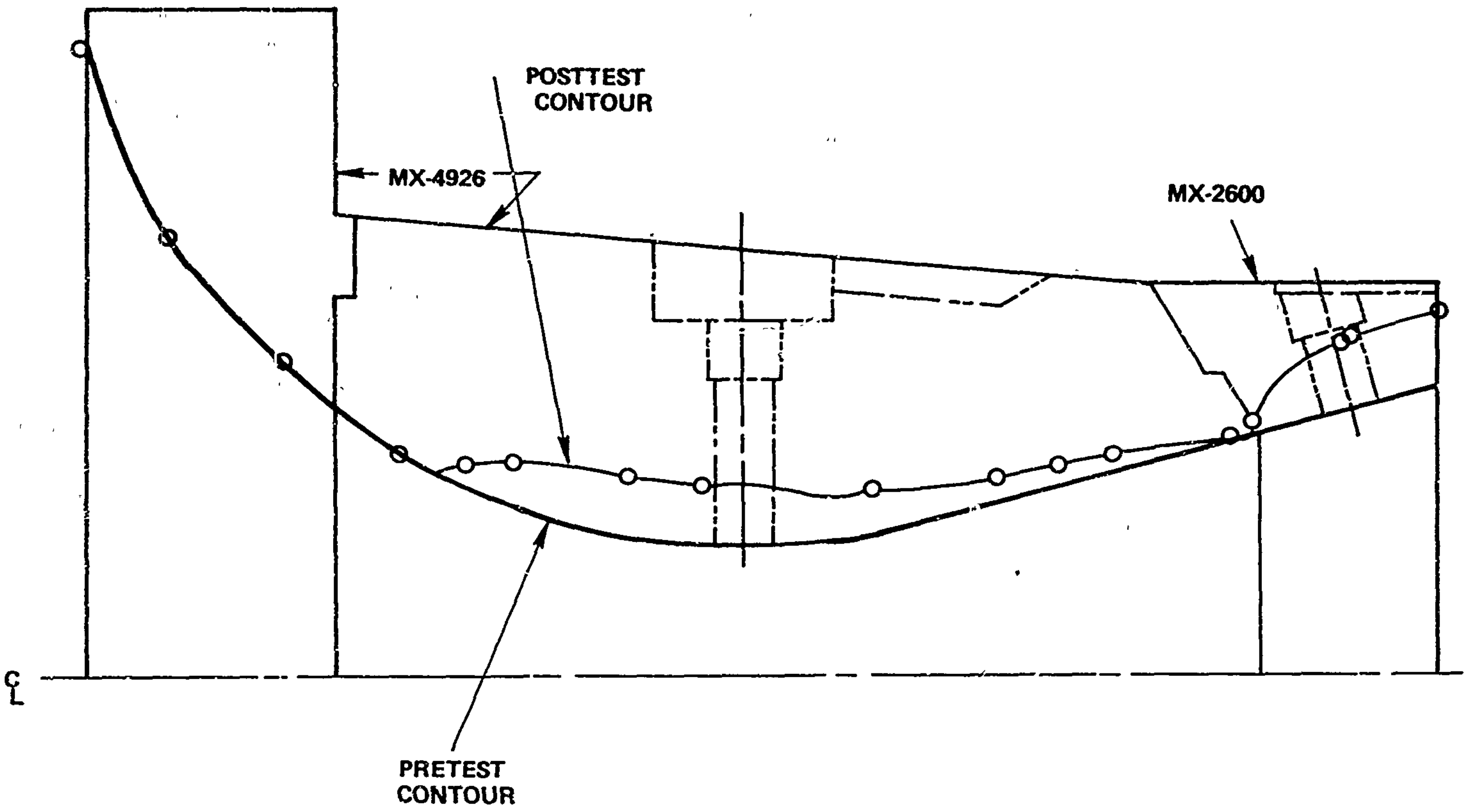

Figure 20. Pretest and Posttest Contour

REFERENCES

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13. ABSTRACT

A solid rocket nozzle instrumented with total and radiative heat-flux transducers was test-fired ~~13 August 1~~969 on the AFRPL 36-inch Char motor. The nozzle was a conventional heavyweight test configuration. The test objective was to measure the total and radiative heat-flux components incident on the nozzle ablative liners. Thermocouple instrumented plugs of ablative material comprised the total heat-flux measurement system. Plugs were placed in the entrance cap, throat, throat extension, and exit sections of the nozzle. A narrow-view angle radiometer provided radiative heat-flux measurements at the throat. A 6500°F flame temperature uncured propellant was used to provide a 1061-psig maximum chamber pressure for a 54-second total firing duration. A description of the heat-flux measurement system concept, nozzle design, motor preparation, performance, and posttest analysis is included in the report.

14. KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
Static test						
Solid propellant rocket						
Char motor						
Ablative materials						
Radiative heat flux						
Total heat flux						
Heat-flux measurement						
High-temperature measurement						
Radiometer						